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INVESTIGATION OF MEANS FOR PERTURBING THE FLOW FIELD IN A SUPERSONIC WIND TUNNEL

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SUMMARY

During the past several years, an attempt has been made at Lewis Research Center to develop a device for perturbing the flow field in a supersonic wind tunnel. The goal of this work was to generate atmospheric type disturbances (e.g., gusts) and to investigate their effects on the dynamic and controls of supersonic inlets. Experimental data were also needed for verification and/or improvement of a NASA analysis of inlet dynamics for disturbances upstream of the normal shock. This report summarizes the status of development of a disturbance device including the desired aerodynamic and actuation capabilities of the device, and the techniques that have been considered and their drawbacks. At the present time no device has been found that satisfies the desired capabilities.

E-9703

INTRODUCTION

Propulsion systems for advanced supersonic cruise aircraft are being identified by studies for the NASA SCAR program. Generally, these systems utilize a mixed-compression type inlet for high performance. A subject of major concern for these inlets is the problem of inlet unstarts induced by atmospheric disturbances. This is a subject for which relatively little information is available from analytical studies or from wind tunnel and flight programs.

To provide a greater understanding of this problem, a flight program and analytical studies have been conducted at Dryden Flight Research Center (DFRC) and Lewis Research Center (LeRC). At DFRC a YF-12 aircraft with an instrumented inlet and a gust sensing probe has been flown. However, the results from program have been limited. Very few encounters with fast, large-amplitude disturbances were recorded, and the effects of these on the inlet appear to be negligible, indicating that the forebody may wash out the effects of turbulence.

An analytical study of atmospheric effects on mixed-compression inlets was recently conducted at LeRC. The study was based on a linear, 1-dimensional mathematical analysis that was derived at LeRC in 1968 (ref. 1). The analysis had been used only to investigate controls for internal (engine induced) disturbances until about two years ago, when it was extended to the external (atmospherically-induced) disturbance problem (ref. 2). Prior to the recent study, the analysis was modified for a significant geometric nonlinearity to more realistically simulate inlet-normal-shock dynamic-behavior in the vicinity of unstart. The results of the study (ref. 3) indicate that inlet control requirements may well be set by rapid atmospheric disturbances. The reason is that inlet normal-shock response to rapid atmospheric disturbances is not attenuated like it is for rapid engine disturbances.

Since the inlet response to atmospheric induced disturbances is so different from the response to engine induced disturbances, it is desirable to be able to investigate the atmospheric cases in the wind tunnel. The analysis can provide a useful tool for minimizing tunnel running time by narrowing down the choice of control concepts to be investigated. This was done for the engine disturbance case (e.g., ref. 4). However, very little data exists for verification of the analysis for upstream disturbances. To provide such data requires a device that produces a relatively uniform disturbance of the tunnel flowfield. Attempts have been made to do this in the 10- by 10-foot Supersonic Wind Tunnel without great success. The devices that have been tried are: (1) crude falling plates to change tunnel throat area and (2) a servo-driven wing (flat trapezoidal-shaped plate) in the test section. The falling plate does not produce the desired waveform nor a uniform change in flow-field properties. The wing had inadequate frequency response (good to about 10 Hz) and did not give the desired change in flow-field properties. This report describes the aerodynamic and actuation capabilities desired for a disturbance device, and describes several devices that have been considered to date and their drawbacks.

SYMBOLS

f_{req}	upper limit of disturbance frequencies required for disturbance device, Hz
L	length of inlet subsonic duct from throat to diffuser exit, cm
M	Mach number
P_t	stagnation pressure, N/cm^2
p	static pressure, N/cm^2
Re	Reynolds number per meter, m^{-1}
r_c	inlet capture radius, cm
T_t	stagnation temperature, K
T	static temperature, K
u	relative air speed, m/sec
W_c	engine corrected airflow, kg/sec
x_s	normal shock position, cm
α	flat plate angle of attack, deg
Δ	change from steady-state value
$\tilde{\delta}$	perturbation variable $\Delta(\,)/(\bar{\ })$

Subscripts:

0	free-stream or ambient value

Superscripts:

$(\bar{\ })$	steady-state value of ( )

DESIRED DISTURBANCE-DEVICE CAPABILITIES

The desired capabilities of the disturbance device must be defined in two major areas - The aerodynamic perturbations it can produce and the speed of response with which the device can be actuated. Aerodynamic capabilities will be considered first.

The major atmospheric variables that can be perturbed and that are easily determined from wind tunnel measurements are total pressure and temperature and Mach number. The relationships between static and total temperatures and pressures Mach number and relative air speed are as follows:

$$M = \frac{u}{65.74\sqrt{T}} \tag{1}$$

$$\frac{p}{P_t} = \left(1 + 0.2M^2\right)^{-7/2} \tag{2}$$

$$\frac{T}{T_t} = \left(1 + 0.2M^2\right)^{-1} \tag{3}$$

Linearization of equations (1), (2), and (3) yields the following set of equations:

$$\frac{\Delta M}{\overline{M}} = \frac{\Delta u}{\overline{u}} - \frac{1}{2}\frac{\Delta T}{\overline{T}} \tag{4}$$

$$\frac{\Delta P_t}{\overline{P}_t} = \frac{\Delta p}{\overline{p}} + \frac{1.4\overline{M}^2}{1 + 0.2\overline{M}^2}\frac{\Delta M}{\overline{M}} \tag{5}$$

$$\frac{\Delta T_t}{\overline{T}_t} = \frac{\Delta T}{\overline{T}} + \frac{0.4\overline{M}^2}{1 + 0.2\overline{M}^2}\frac{\Delta M}{\overline{M}} \tag{6}$$

Equation (4) indicates that a decrease in free-stream Mach number can result only from an increase in static (ambient) temperature or from a decrease in relative air speed. For independent perturbations in relative air speed, ambient pressure and ambient temperature, equations (4) to (6) reduce to the following sets of equations:

Relative air speed perturbation, $\Delta u_0/u_0$

$$\frac{\Delta M_0}{\overline{M}_0} = \frac{\Delta u_0}{\overline{u}_0} \tag{7}$$

$$\frac{\Delta P_{t0}}{\overline{P}_{t0}} = \frac{1.4\overline{M}_0^2}{1 + 0.2\overline{M}_0^2} \frac{\Delta u_0}{\overline{u}_0} \tag{8}$$

$$\frac{\Delta T_{t0}}{\overline{T}_{t0}} = \frac{0.4\overline{M}_0^2}{1 + 0.2\overline{M}_0^2} \frac{\Delta u_0}{\overline{u}_0} \tag{9}$$

Ambient pressure perturbation, $\Delta p_0/\overline{p}_0$

$$\frac{\Delta M_0}{\overline{M}_0} = 0 \tag{10}$$

$$\frac{\Delta P_{t0}}{\overline{P}_{t0}} = \frac{\Delta p_0}{\overline{p}_0} \tag{11}$$

$$\frac{\Delta T_{t0}}{\overline{T}_{t0}} = 0 \tag{12}$$

Ambient temperature perturbation, $\Delta T_0/\overline{T}_0$

$$\frac{\Delta M_0}{\overline{M}_0} = -\frac{1}{2}\frac{\Delta T_0}{\overline{T}_0} \tag{13}$$

$$\frac{\Delta P_{t0}}{\overline{P}_{t0}} = -\frac{0.7\overline{M}_0^2}{1 + 0.2\overline{M}_0^2} \frac{\Delta T_0}{\overline{T}_0} \tag{14}$$

$$\frac{\Delta T_{t0}}{\overline{T}_{t0}} = \frac{1}{1 + 0.2M_0^2} \frac{\Delta T_0}{\overline{T}_0} \tag{15}$$

Equations (8), (11), and (14) indicate that free-stream total pressure decreases when either Mach number or ambient pressure decreases. However, total temperature could increase or decrease due to a decrease in Mach number, depending on whether ambient temperature increases or relative air speed decreases.

Some additional observation can be made by knowing how an inlet will respond to perturbations in atmospheric variables. The analysis of reference 2 was applied to a typical mixed-compression inlet configuration (ref. 4) to predict the frequency response of the inlet's normal shock to independent perturbations in ambient pressure, ambient temperature and relative velocity. The results are shown in figure 1 where the gain of shock position to each perturbation variable is plotted as a function of perturbation frequency. Note that for these perturbations the shock amplitude exhibits a rising trend with frequency up to 40 or 50 hertz and then decreases. This response can be interpreted to mean that the disturbance amplitude required to unstart the inlet decreases as frequency increases up to 40 or 50 hertz. This is an undesirable characteristic and is opposite that found to occur for engine airflow disturbances (ref. 4). Figure 1 indicates that the response of shock position is proportional to the derivative of ambient pressure, (and, hence, total pressure) at low frequencies. The steady-state gain (gain at 0 frequency) is not exactly zero because of some bleed terms. It is the derivative effect that causes all of the responses of figure 1 to exhibit a rising characteristic. Figure 1 also indicates that a change in Mach number due to a change in ambient temperature or relative velocity, produces a significant steady-state change in shock position. The conclusion is that to produce inlet responses in a wind tunnel, having steady-state and dynamic characteristics like those of figure 1, the disturbance device must be capable of producing a simultaneous decrease in both Mach number and total pressure. For example, the sudden appearance of a shock wave in front of the inlet could produce such a combination.

Proper testing in wind tunnels of the effectiveness of throat-bypass stability systems against upstream disturtances also requires that the correct combination of change in free-stream Mach number and total pressure be provided. Generally these systems use passive valves (e.g., relief-type mechanical valves or vortex valves) that bleed airflow out of the inlet when actuated by an increase in inlet-throat pressure. It can be shown that inlet throat static pressure will generally increase when inlet free stream Mach number decreases provided inlet geometry remains fixed. However the magnitude of the pressure increase can be very small or quite large, depending on the relative changes in free-stream Mach number and total pressure. Hence, when wind tunnel tests show that a certain decrease in free-stream Mach number is sufficient to actuate a stability system, proper bookkeeping must be done to be sure that the same (or at least a sufficient) increase in actuating pressure would occur for the same decrease in Mach number in a flight environment. For example, disturbances produced in the 10- by 10-foot Supersonic Wind Tunnel by changing tunnel throat blockage have resulted in an increase in free-stream total pressure with a decrease in free-stream Mach number. This would result in an increase in inlet-throat-static pressure that would be unrealistically higher than the increase that would occur in flight.

For inlet testing in the wind tunnel it would also be desirable for the gain of shock position to the disturbance to be large so that the disturbance amplitude can be relatively small. For the inlet of figure 1 the gain of shock position to engine corrected airflow $(\Delta x_s/r_c / \Delta W_c/\overline{W}_c)$ is 6.669, where a negative Δx_s is in the upstream direction toward unstart. Based on that gain and gains from figure 1 it can be shown that a steady-state reduction in Mach number of 0.026 would be required to unstart the inlet when the shock is positioned to have an engine corrected airflow margin from unstart of 1 percent. This assumes that the normal shock distance from unstart is the same for both external and internal disturbances. A typical corrected airflow margin from unstart is on the order of 3 to 5 percent necessitating a steady-state change in Mach number of 0.078 to 0.13 to unstart the inlet. An inlet with a more effective throat bleed would require a greater reduction in Mach number to unstart. Taking these numbers as values for a typical inlet it appears that the disturbance device should be capable of producing a steady-state change in tunnel test section Mach number of about 0.1. It can be shown from equation (5) that the corresponding change in total pressure ($\Delta P_t/\overline{P}_t$) would be 0.1555 assuming that no change in ambient pressure occurs and that the initial Mach number $\overline{M}_0$ is 2.5. In actual practice it may not be possible to attain these magnitudes. More realistic values would probably be a change in Mach number of 0.05 and a corresponding change in pressure of 0.0778. The disturbance amplitude decreases as disturbance frequency increases, as mentioned previously.

The discussion above should be qualified to the following extent. It is doubtful that perturbations in ambient temperature, ambient pressure and relative velocity (gusts) occur independently in the atmosphere. At present it is not clear that perturbations occur in any prescribed manner. However it does seem certain that a reduction in Mach number would normally be accompanied by a reduction in total pressure.

Speed of response capabilities required by the disturbance device may also be deduced from figure 1. It would be desirable to be able to define the first resonant peak that occurs in the range of 40 to 50 hertz. For good characterization of the peak, the device would have to be capable of frequencies in the vicinity of 60 to 70 hertz. The frequency requirements will be influenced by model size and tunnel total temperature T_{t0} in the following manner:

$$f_{req} \sim \frac{\sqrt{T_{t0}}}{L} \tag{16}$$

where L is the length of the subonic duct. For inlets scaled up or down from the one modeled in figure 1 the disturbance device frequency response requirements become (worst case):

$$f_{req} = (70)\sqrt{\frac{T_{t0}}{313.3}}\ \frac{23.7}{r_c}\ \text{Hz} \tag{17}$$

where r_c is the capture radius of the inlet. Thus frequency requirements can be reduced by increasing model size and decreasing tunnel total temperature. In the 10- by 10-foot Supersonic Wind Tunnel, for example, the maximum allowable cowl radius is about 76 centimeters and the minimum temperature about 303.9 K (at $M = 2.5$ and $Re = 1.64\times10^6$ per meter). The required frequency f_{req} then becomes 21.5 hertz. The disadvantage to increasing model size is that the uniform flow field to be perturbed must be correspondingly larger.

A few words about actuation requirements is appropriate at this point. The capability to produce a pulse-type disturbance (e.g., triangular wave form) is desirable. The greatest flexibility would result from being able to adjust both amplitude and duration in a continuous manner. Although a pulse testing technique is desired, it may not be possible to build a servo-driven device that can produce the required short duration pulse (high frequency) disturbances. An alternate approach would be to use motors and cams to drive the device sinusoidally or in some cyclical manner but with a constant amplitude. In that case, an alternate test procedure, similar to Wasserbauer's unstart method (ref. 5), could be used. The test procedure would be to adjust diffuser exit (engine) corrected airflow to the lowest possible value without unstart at each disturbance frequency. The unstart method or the pulse testing could be duplicated using the inlet simulation. The unstart method is probably more efficient in terms of test time required, but has the disadvantage that the effect of operating point is not easily determined.

The required disturbance device capabilities outlined above are based on the assumption that the atmospheric disturbances occur in nature in a manner similar to that assumed for figure 1 and that the results predicted by the LeRC analysis are correct. Although the rising characteristic predicted by the analysis has not been verified by experiment at LeRC, it should be noted that the analysis does agree in this respect with inlet transfer function models and with results from a method of characteristics solution, all in reference 6. A comparison of the LeRC analysis with results from the method of characteristics solution is shown in figure 2. Phase angle agreement is excellent and the shapes of the amplitude curves also agree quite well. The major discrepancy appears to be in the low frequency gain. The discrepancy is believed to be due to nonlinearities that have a greater effect on the gain for disturbances upstream of the normal shock then for downstream disturbances. It should be possible to improve the steady state gain characteristics of the analysis by the addition of some nonlinear terms in the same manner that the analysis was modified for the geometric nonlinearity mentioned earlier.

DISTURBANCE TECHNIQUES

Four means for inducing a change in test section Mach number were given consideration. They are shown schematically in figure 3, and are listed in order of showing promise as follows:

(1) A change in angle of attack of a triangular airfoil located at the upstream end of the test section, spanning the tunnel walls.

(2) Modulation of tunnel throat area (by flexing sidewalls or by expanding and contracting a centerbody).

(3) Change in angle of attack of a flat plate - similar to the trapezoidal plate used by Wasserbauer (ref. 5).

(4) Blast wave from a shock tube fired into the supersonic stream.

This section describes the relative advantages of the four disturbance techniques, beginning with the least promising.

A blast wave from a shock tube (fig. 3(d)) was considered because it should produce a rapid disturbance. Information was found in the literature (refs. 7 and 8) that described tests in which blast waves were fired in supersonic wind tunnels either perpendicular to or in the same direction as the tunnel flow. Both blasts produce rapid disturbances (perhaps too fast for the inlet application). The research was aimed primarily at measuring overpressures due to the blast wave on bodies traveling at supersonic speeds. The side blast is eliminated from consideration because it does not provide a uniform disturbance. A head-on blast does not produce a good waveform because of shock waves being reflected back upstream from the inlet. Also, control of amplitude and duration would probably be inadequate and a dc or slow disturbance could not be produced. Implementation is simple, however, so this could represent a fall back position for getting some limited data if a more controlled technique failed.

The flat plate (fig. 3(c)) has several disadvantages. First, it produces a change in flow angularity as well as Mach number. The flow angularity cannot be handled by the LeRC 1-dimensional analysis. Also, the change in total pressure caused by small changes in plate angle of attack is negligible. Therefore, the response preducted by the analysis as shown in figure 4 does not show the desired rising characteristic of figure 1. Finally the frequency response of the existing plate is limited to about 10 hertz. The flat plate could be used to provide experimental data for verification, and/or improvement of the LeRC inlet-analysis steady-state characteristics. But there would be little advantage in lightening the plate and using special actuators to drive it out to 20 or 30 hertz because of the lack of a peak in the inlet's response to its disturbance.

The throat modulation technique (fig. 3(b)) has drawbacks similar to those of the flat plate. The analysis predicts that the inlet response to this disturbance would be similar to that due to the flat plate when it is assumed that the disturbance is a pure area or flow disturbance at the throat. (This assumption neglects shock waves that could be generated by such a device.) Unpublished steady-state results from an analytical study indicated that a throat centerbody device will generate shock waves. The shock waves reflect down the tunnel and result in nonuniform flow at the inlet. Also, the total pressure would increase when Mach number decreases because shock strength is reduced with the centerbody collapsed. Another drawback of throat modulation is that the disturbance is not near the inlet. Therefore, to compare experimental results with analysis, either the tunnel must be simulated (as well as the inlet) or careful measurement of the disturbance must be made as a function of frequency (or pulse width) at the inlet location. This problem might not occur, or would be less severe for a device operating in close proximity to the inlet. Actuation and design of a throat device would also be more difficult than for the other disturbance methods discussed. Therefore, throat modulation was eliminated as a prime candidate for testing.

The triangular airfoil device of figure 3(a) appeared to offer the most promise in terms of aerodynamic capability. Results of a simple 2-dimensional flow field analyses, as shown in figure 5, predict that an inlet located in either region 1 or region 2 will be subjected to the desired disturbance (simultaneous decrease in pressure and Mach number) as airfoil angle of attack increases in the range of about 6 to 20 degrees. The LeRC inlet analysis was also exercised to predict inlet normal shock response to perturbations in airfoil angle of attack. The results, shown in figure 6, indicate that the airfoil will produce dynamic characteristics similar to those of figure 1. It would appear obvious that the inlet should be tested in disturbance region 2 because the change in flow-field properties (fig. 5) and shock position (fig. 6) with angle of attack is much greater than for region 1. A triangular airfoil device like the one of figure 3(a) was built by the Lockhead Aircraft Corp. and tests were conducted in the Ames 8- by 7-foot Supersonic Wind Tunnel (ref. 9). During the Ames tests the inlet was located in region 1, below the slip line, rather than in region 2 as shown in figure 3(a). Flow field measurements were made only in region 1. The steady-state experimental results were not promising. The change in flow field properties across the inlet was not uniform with changes in airfoil angle of attack. However, experimental frequency responses of shock position to angle of attack did exhibit the rising characteristic predicted in figure 6 for region 1. The poor steady-state results could have been due to slip line interference with the inlet and due to three dimensional effects from a strut that was used to support and actuate the airfoil. Increasing the chord length of the airfoil would have increased the size of the uniform flow field (but not the strength of disturbance) as will be discussed later.

Because of the poor results for region 1, it was hoped that an inlet could be tested in region 2. However, it was felt that the results of figure 5 could be significantly in error because the effects of the expansion fan on the near-wake flow-field of the airfoil were not accounted for by the simple analysis. Therefore, a two dimensional inlet analysis program, using the method of characteristics, was adapted to the problem. Since this is an inlet analysis program, the airfoil geometry and slip line were included as part of the inlet cowl surface and the centerbody surface was treated as a tunnel boundary (ceiling or floor). The results showed that the expansion nearly washes out the effects of the airfoil in the vicinity of region 2 where a model would be located. The net result is a very weak disturbance more like that of region 1. The results did indicate that increasing the chord length of the airfoil resulted in a larger uniform flow field in both regions 1 and 2, as expected, but did not significantly increase the strength of the disturbance in region 2. Some of the results of this analysis are presented in the appendix.

The detailed analysis also indicated that testing would have to be limited to inlets with a capture diameter of about 50 centimeters or less and that inlet ingestion of shock waves and/or Mach waves generated by the airfoil still could be a problem, especially if a vertical support strut is required to prevent bending. Tunnel-wall boundary-layer separation due to impingement of shock waves generated by the airfoil could also be a problem. Finally the airfoil could produce a change in Mach number of only 0.025 without causing a change in flow field angularity of more than 1 degree. Due to all of these considerations the airfoil was eliminated as a disturbance device candidate.

CONCLUSIONS

It was found that a device for simulating atmospheric type disturbances in supersonic wind tunnels should be capable of producing relatively uniform flow field perturbations with very small or no change in angle of attack and with a simultaneous decrease in Mach number and total pressure (on the order of at least 0.05 and 8 percent, respectively). Actuation should be devised such that the flow field perturbation can be a single pulse (e.g., triangular wave) and/or cyclical. Disturbance frequency requirements can be decreased by increasing inlet size and decreasing tunnel total temperature. However, increasing inlet size has the disadvantage that the uniform flow field to be perturbed must also be larger. No device has been found that satisfies the desired capabilities at the present time.

APPENDIX - DISCUSSION OF DETAILED FLOW-FIELD ANALYSIS AND RESULTS

A detailed analysis was used to investigate the flow field of disturbance device configurations with cross sections like the shaded shapes shown in figure 7. The shock waves (solid lines), expansion fans (dashed lines with included angle), and slip lines are shown schematically. The shock waves would actually curve because of interaction with the expansion fans. This effect was ignored in the simple analysis. Detailed flow field calculations were made by adapting the disturbance device geometry to an analysis that computes two-dimensional flow in supersonic inlets using a weakly viscous method of characteristics. The disturbance device geometry and trailing edge slip line were treated as the internal-cowl surface and the tunnel boundary served as the inlet centerbody surface. Mach numbers computed along each configuration's surface and the slip line right at the trailing edge agreed almost exactly with the simple analysis, as expected. An example of how the geometry is adapted for calculation of triangular airfoil disturbance region 2 is shown in figure 8. The centerbody was positioned so that its tip shock would fall upstream of the cowl lip (airfoil leading edge). The analysis treats the centerbody-tip shock and the cowl-lip shock and reflections explicitly. However, imbedded shocks, like the one at the trailing edge are smeared and associated total pressure losses are not accounted for. A drawback of using the detailed analysis is that the flow field above the flat side of the airfoil and slip line is ignored because the slip line must be treated as a solid boundary. The slip line is assumed to extend straight from the trailing edge at the angle determined by the simple analysis. This is correct for conditions right at the trailing edge, but for conditions further downstream the slip line would actually curve due to interaction of the flow fields above and below the surfaces of the airfoil and slip line. By neglecting this effect, the detailed analysis probably showed a greater change of Mach number in region 2, then would actually occur. The detailed analysis also ignores the possibility of boundary layer separations and the finite thickness of the slip line, as does the simple analysis. Additional details concerning the analysis are presented in reference 10.

The strut with symmetrical trailing-edge flap, configuration b of figure 7, was investigated because it offered some advantages in terms of tunnel installation. It was determined that a vertical support strut, required to prevent bending of the triangular airfoil, would not be needed for the strut-flap combination. Besides being an installation advantage, it also eliminates a source of three-dimensional flow field effects. The strut-flap combination was eliminated from consideration later, however. The reason is that the inlet ingestion of the leading edge reflected shock became a problem when flap chord length had to be increased to provide a uniform flow field large enough for the inlet size of figure 1 (47.4 cm capture diameter). Therefore, the remainder of the appendix will be confined to a discussion of the application of the detailed analysis to the triangular airfoil, configuration a of figure 7.

The detailed analysis was initially applied to an airfoil with an 20.3 centimeter chord length. It was found that this airfoil did not produce a uniform flow field in regions 1 or 2 that was large enough to accommodate the 47.4 centimeter diameter inlet. The size of the uniform flow region can be increased by increasing the chord length of the airfoil, as suggested by force-momentum considerations of control volumes and by the delayed intersection of the expansion waves with the shock waves. Increasing the chord length would make the airfoil more difficult to actuate at high frequencies, but could increase stiffness enough to eliminate the need for a support strut to prevent bending due to lift and drag. The alternative to increasing chord length is to test an inlet with a smaller capture diameter. However, that increases the disturbance frequency requirements as indicated by equation 17.

The detailed analysis was used to compute the flow field generated by the flat side of the airfoil. The analysis determines flow field properties between the tunnel boundary (floor or ceiling) and the slip line at various longitudinal locations. Mach number profiles at the expected longitudinal location of the inlet cowl lip, as predicted from the simple analysis, are shown in figure 9 for two different chord lengths. Local

Mach number is plotted as a function of distance perpendicular to the tunnel floor. (The normalizing length has no special significance to the problem and because of the coordinate system used, the floor is not at a value of 0.) The profiles exhibit sudden decreases and gradual increases in Mach number that are characteristic of shock compressions and expansion fields respectively. The portion of the profiles that is of major interest is for region 1. It occurs at normalized distance values in the range of about 0.835 to 1.045. Note that in region 1 the constant Mach number region of 2.586 for the 61.0 centimeter chord airfoil is more than twice as big as that for the 20.3 centimeter airfoil. The results for the 20.3 centimeter airfoil exhibit an overexpansion to about Mach 2.6 before recompressing to Mach 2.586. Experimental data, given in reference 9 for a 20.3 centimeter airfoil under the same conditions, exhibited a similar overexpansion characteristic that was even more extreme (the maximum Mach number being 2.65). The Mach number of 2.586 predicted by the detailed analysis for the uniform region agreed exactly with that predicted by the simple analysis. But the simple analysis predicted a constant Mach number flow field for the 20.3 centimeter airfoil that was about the same size as that predicted by the detailed analysis for the 61.0 centimeter airfoil. The detailed analysis also predicted that the larger chord airfoil would produce a larger uniform field in region 2 than the smaller chord airfoil, although not large enough to accommodate the 47.4 centimeter diameter inlet. Clearly, increasing chord length is beneficial from an aerodynamic standpoint.

A major impetus for applying the detailed analysis was to determine the effects of the expansion around the upper surface of the airfoil (fig. 7(a)) on the flow field in region 2. Cases were run for a 61.0 centimeter chord airfoil at two angles of attack. The expansion corner was smoothed to ease the computational process but the turning angle remained the same. Mach number profiles at the expected longitudinal location of the inlet cowl lip, as predicted from the simple analysis, are shown in figure 10. In these cases the profile is plotted between the tunnel ceiling (at the bottom of the plot) and the slip line boundary. Again the profiles exhibit the sudden compression and gradual expansion characteristics. The portion of the profiles that is of major interest is for region 2. It occurs at normalized distance values in the range of about 0.88 to 1.185. It is apparent that the change in Mach number is not uniform over the 47.4 centimeter capture diameter of the inlet.

The simple analysis, which neglected the expansion effects on the shock waves, had predicted that 20.3 centimeter chord airfoil would produce the desired uniform flow field and a much greater change in Mach number (about 0.08), as indicated in the upper left hand corner of figure 10. The higher absolute Mach numbers predicted by the detailed analysis for region 2 indicate that the expansion field does largely wash out the disturbance.

Based on the results of the detailed flow field analysis and their comparison to the simple analysis results, the following conclusions are drawn:

(1) The simple analysis was adequate for predicting the Mach number in disturbance region 1 but not the size of the uniform flow field.

(2) The expansion field effects, neglected in the simple analysis, are significant and nearly wash out the disturbance that the simple analysis predicts for region 2.

(3) The size of the uniform flow field generated in region 1 is greater than that in region 2 and the magnitude of the disturbance is about the same for both regions.

(4) Increasing the size (chord) of the airfoil increases the size of the uniform flow field, as expected.

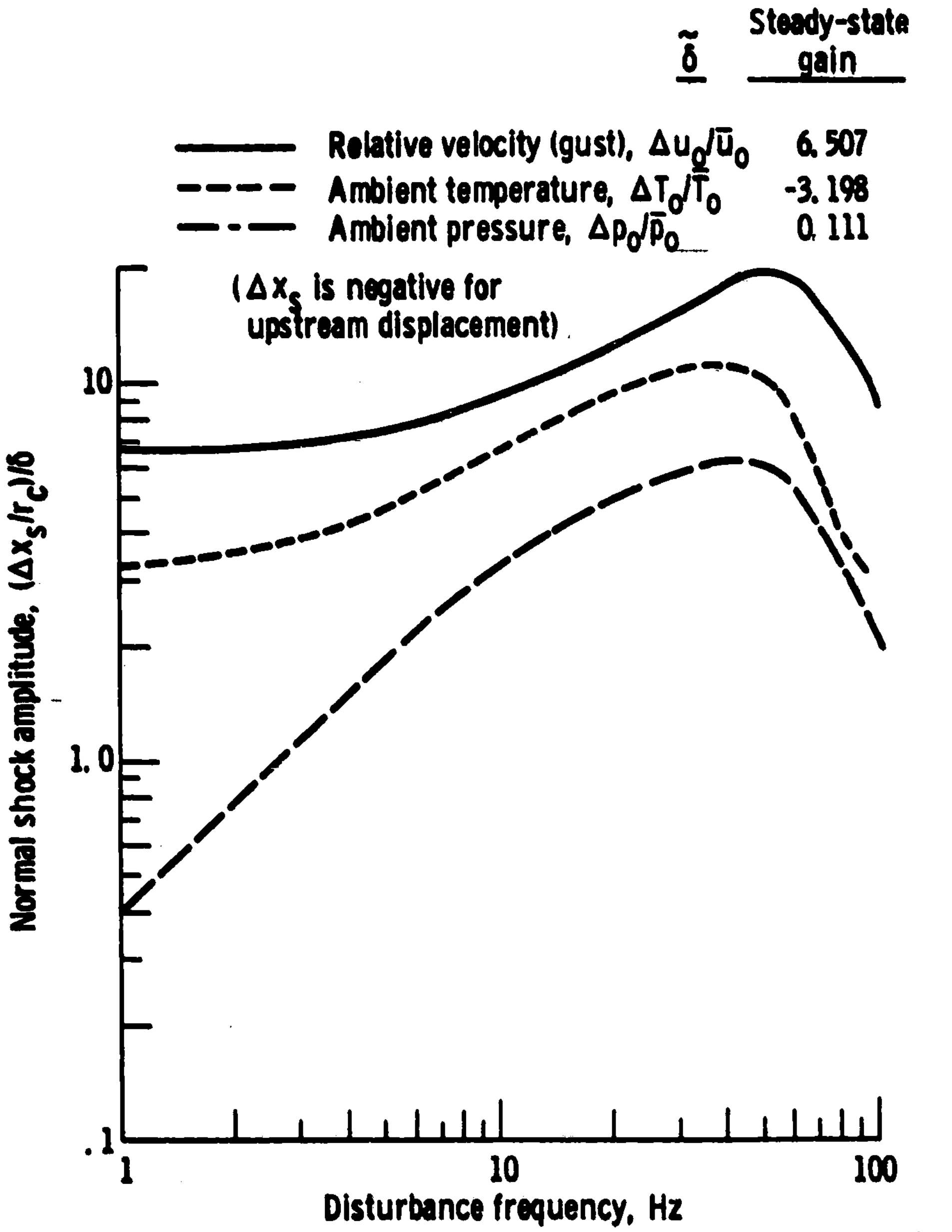

Figure 1. - Response of inlet to independent perturbations of atmospheric variables (without control), as predicted by analysis of ref. 2. Diffuser exit choked. Capture radius, r_c = 23.7 cm; length of subonic duct, 110.5 cm; free-stream Mach number, $\bar{M}_0$ = 2.5; free-stream total temperature, T_{to} = 313.3 K.

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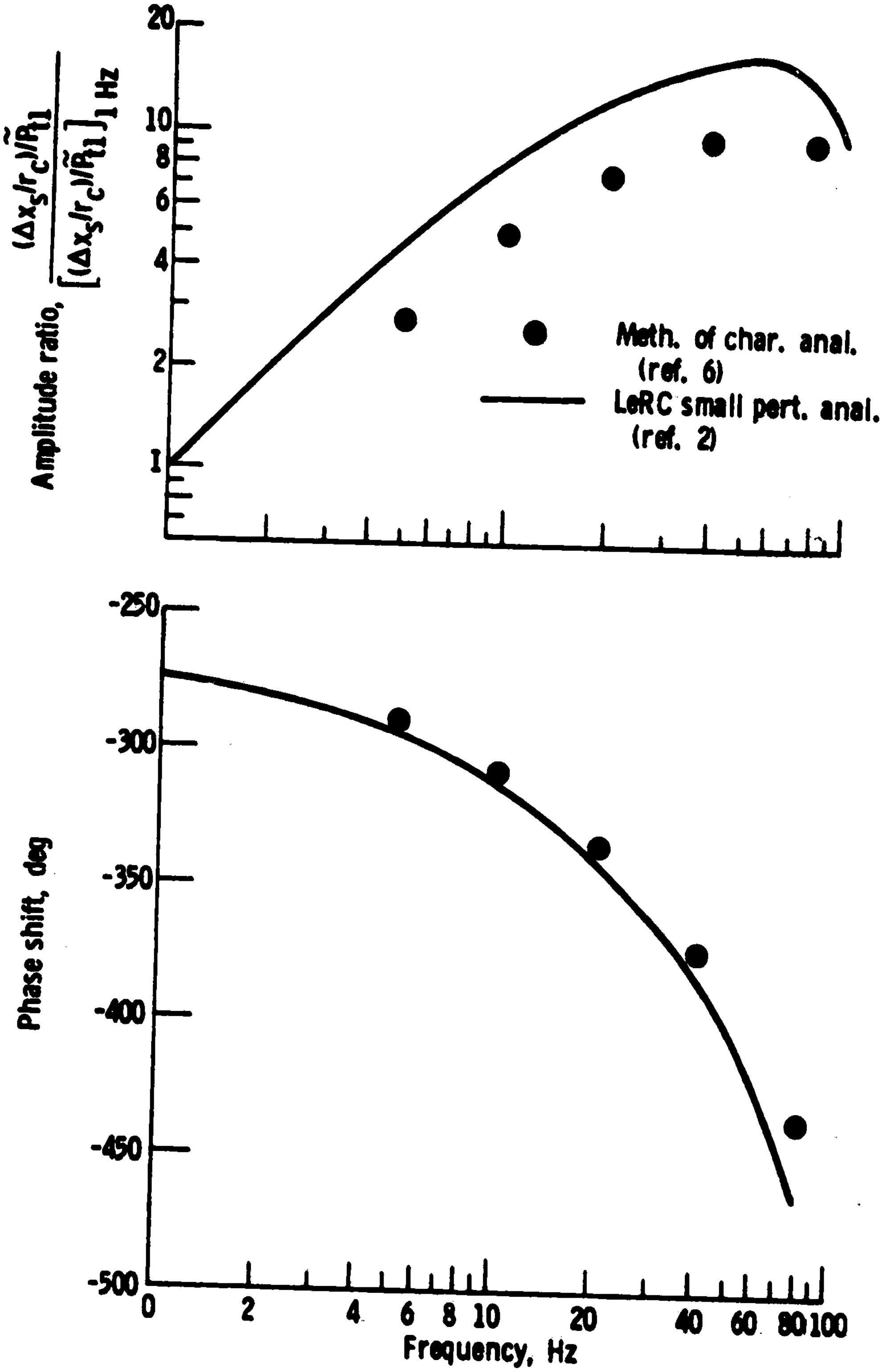


Figure 2. - Comparison of response of inlet normal shock to total pressure perturbation $\tilde{P}_{t1}$ upstream of shock, as determined by method of characteristics and LeRC small perturbation analyses.

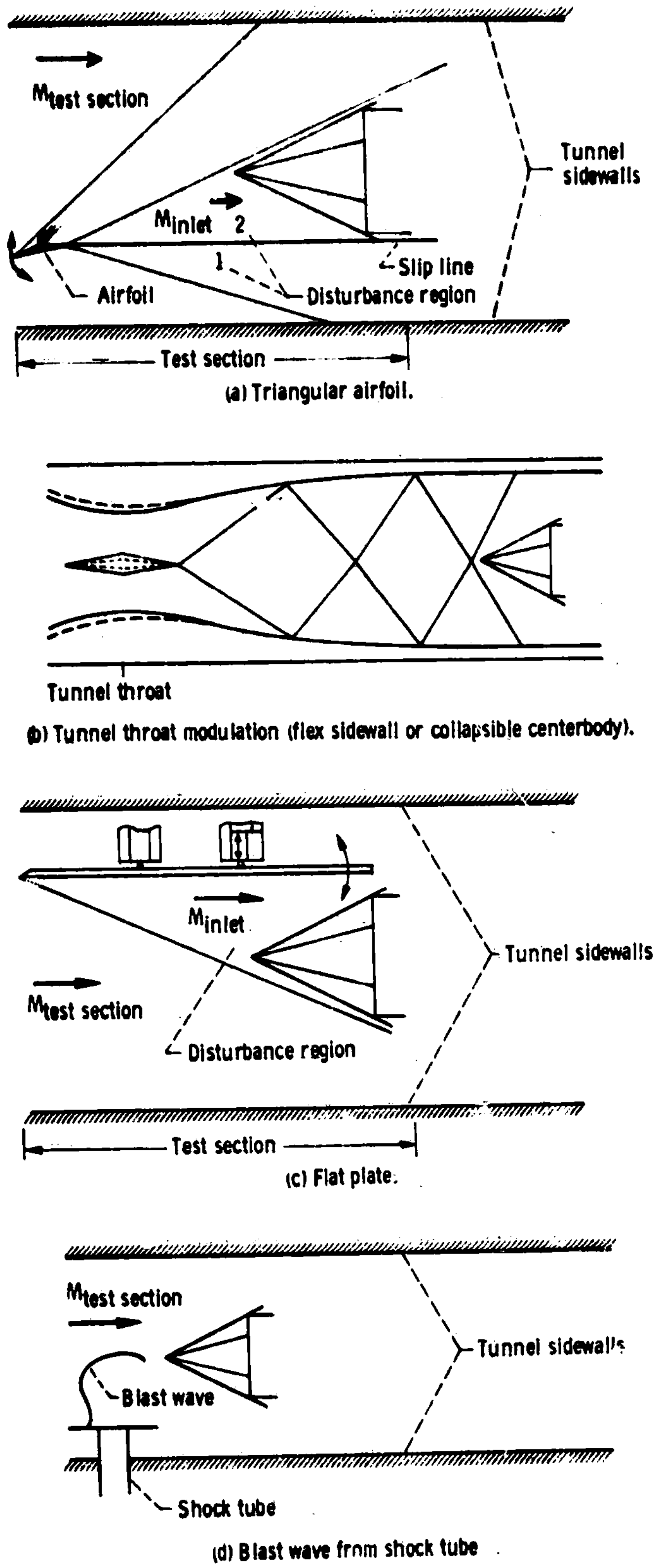


Figure 3. - Schematic of techniques for disturbing the flow field of supersonic wind tunnels.

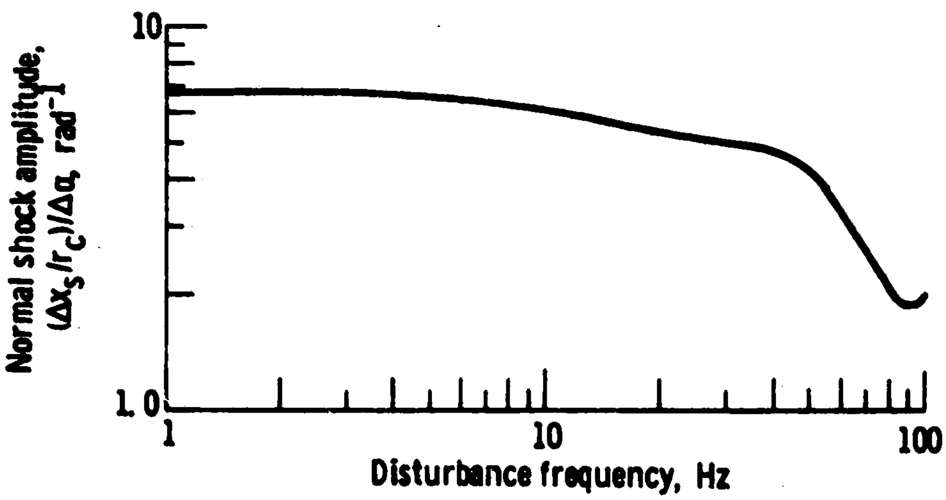


Figure 4. - Response of inlet to angle-of-attack perturbations of flat plate as predicted by analysis of ref. 2. Steady-state gain, $(\Delta x_s/r_c)/\Delta\alpha = 6.75$ rad^{-1} = 2.75 Mach^{-1}. Same inlet configuration and operating point conditions as for fig. 1.

Symbols:

M - Mach no.
PR - P_1/P_0 - P_2/P_0
P - static pressure
TR10 - T_1/T_0
TR20 - T_2/T_0
T - static temp.
σ - slip line angle (flow angularity)
α - airfoil angle of attack
0 - free-stream
1 - region ①
2 - region ②

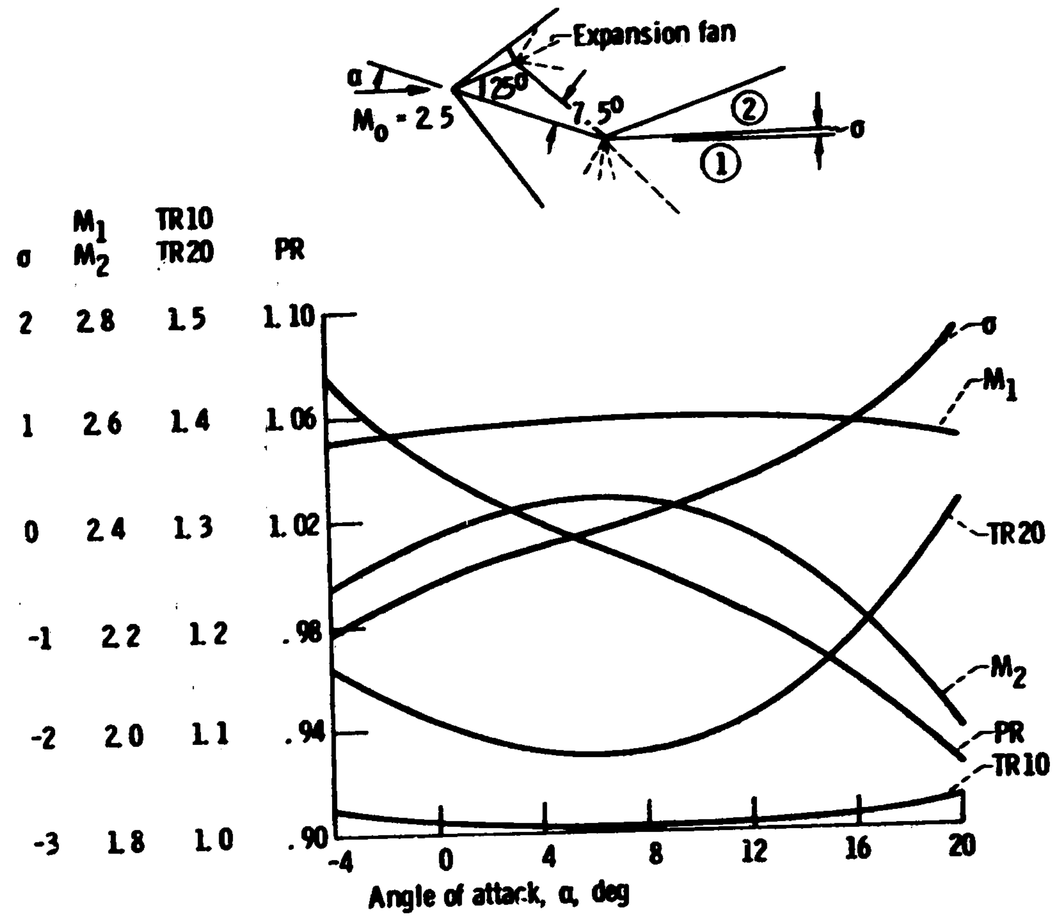


Figure 5. - Flow-field properties in the near wake of a triangular airfoil.

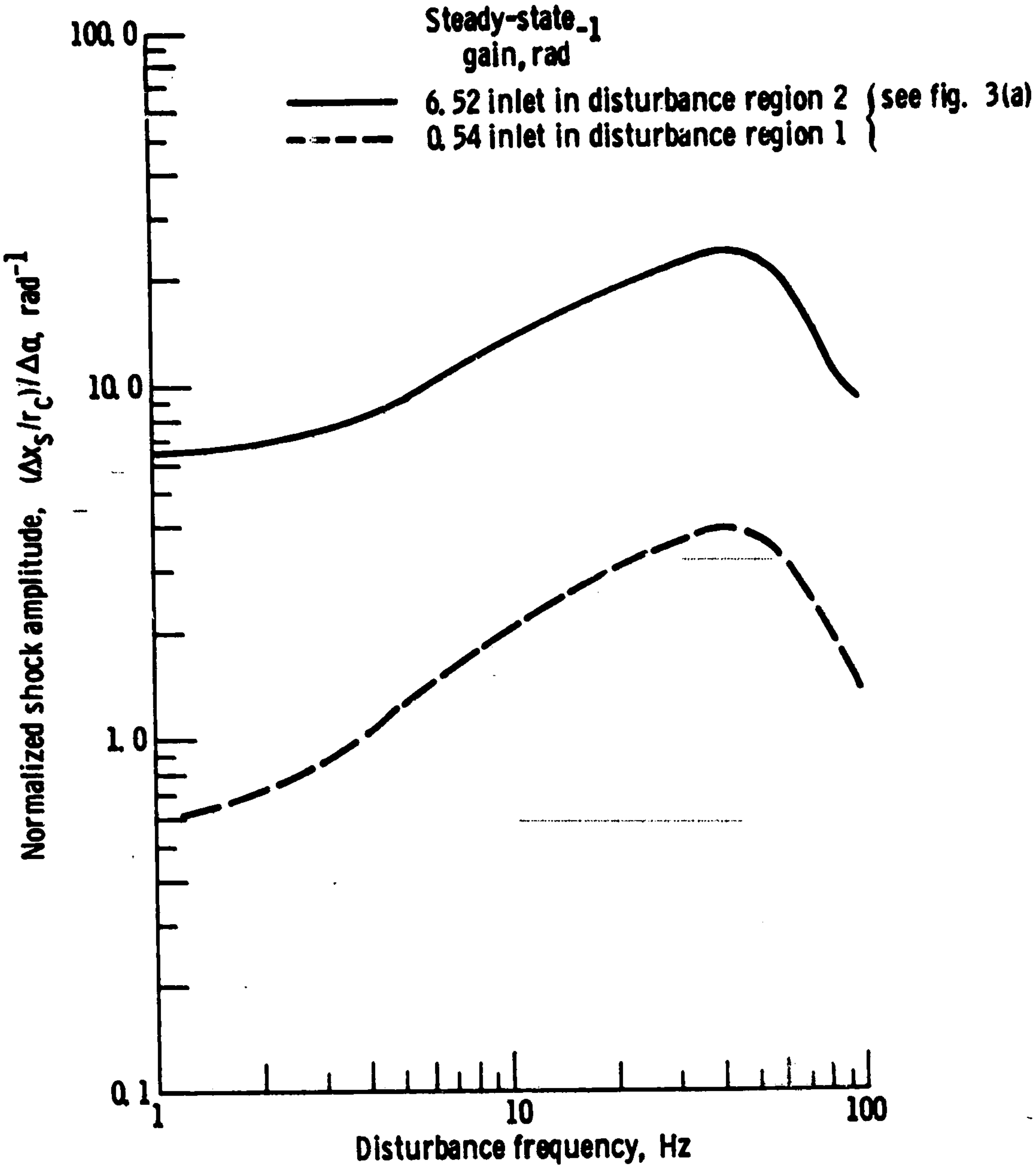
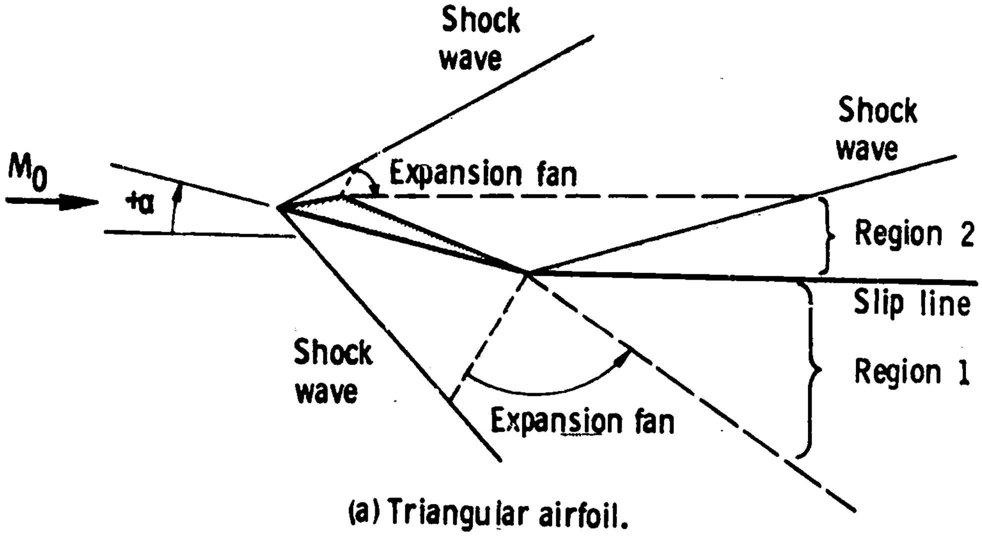
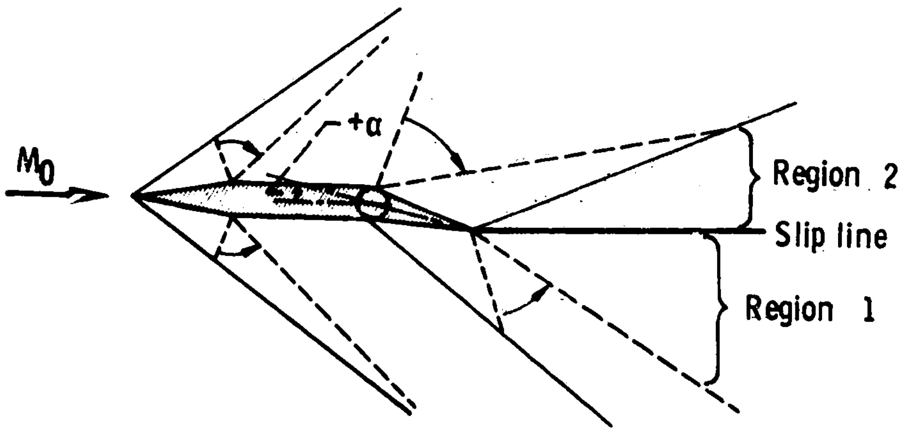


Figure 6. - Response of inlet to perturbations in triangular-airfoil angle of attack ($14^0 \pm 2^0$) as predicted by analysis of ref. 2. Same inlet configuration and operating point conditions as for fig. 1. Triangular airfoil leading edge angle, 25^0; trailing edge angle, 7.5^0.



(a) Triangular airfoil.



(b) Strut with trailing edge flap.

Figure 7. - Disturbance generator configurations investigated with detailed flow-field analysis.

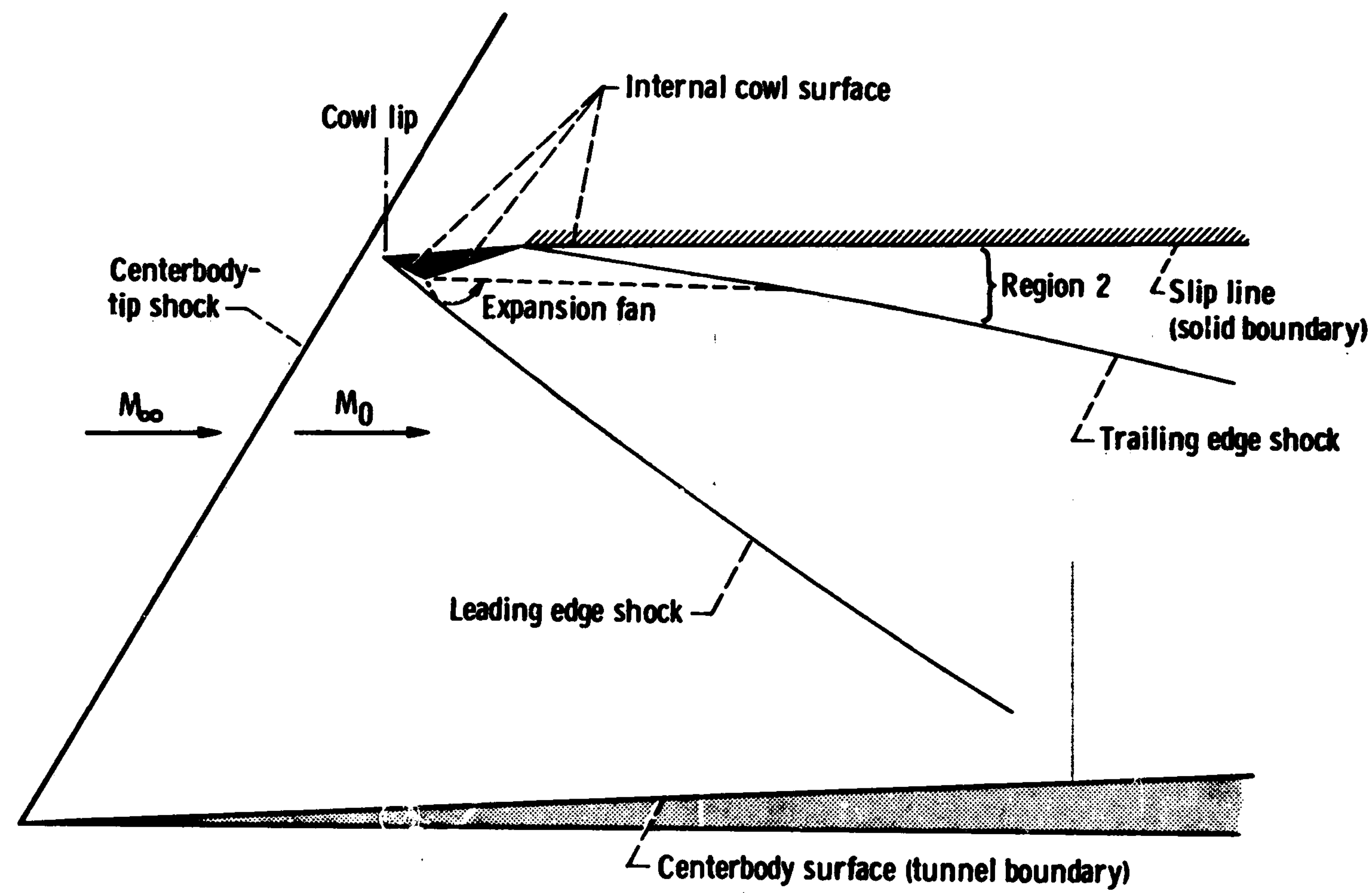


Figure 8. - Typical disturbance-generator/wind-tunnel geometry adaptation to 2-dimensional inlet analysis.

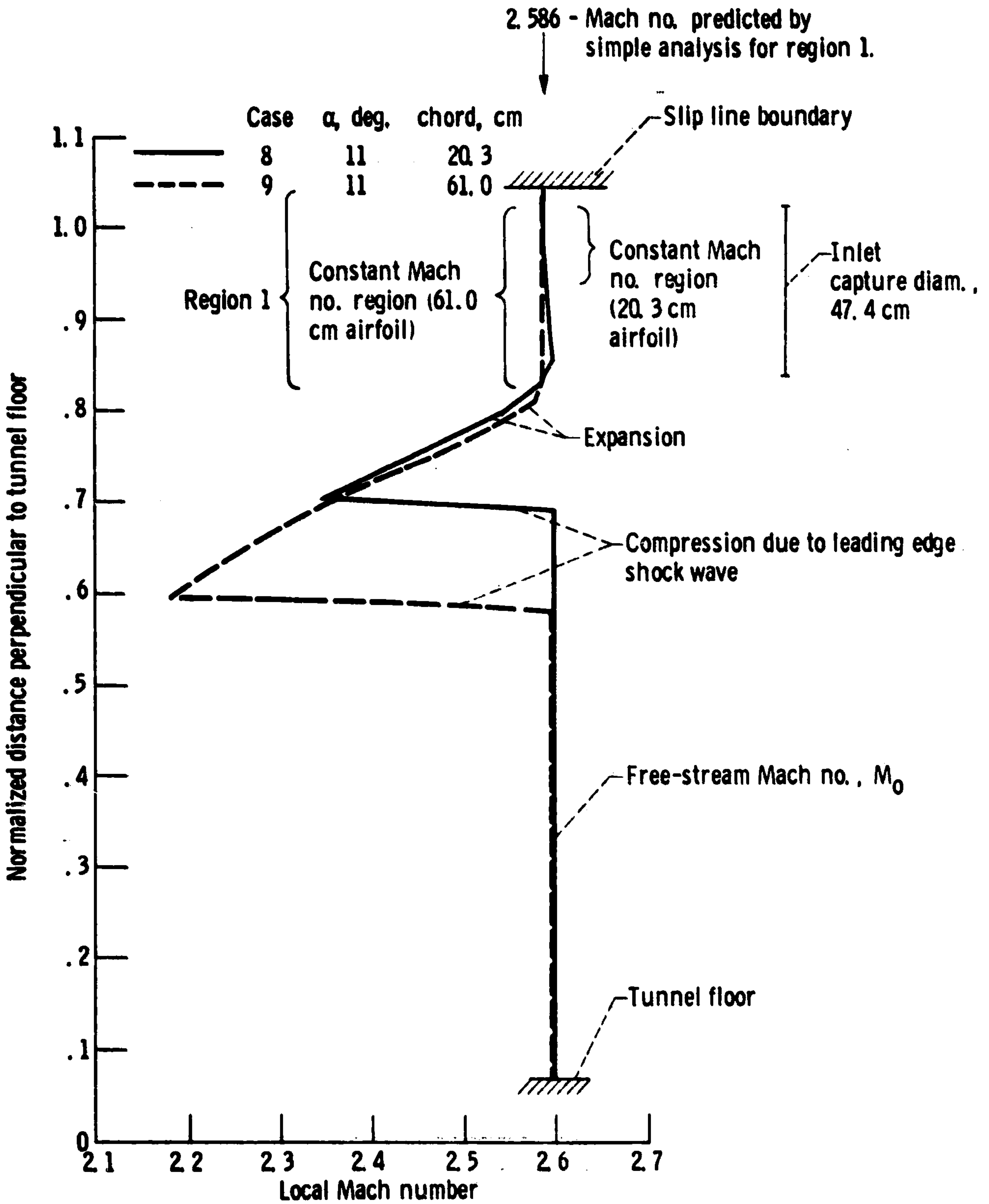


Figure 9. - Mach no. profiles predicted by detailed analysis at expected longitudinal location of inlet cowl lip. Triangular airfoil disturbance (see fig. 7). Determination of disturance region 1. Normalizing length, 260.1 cm.

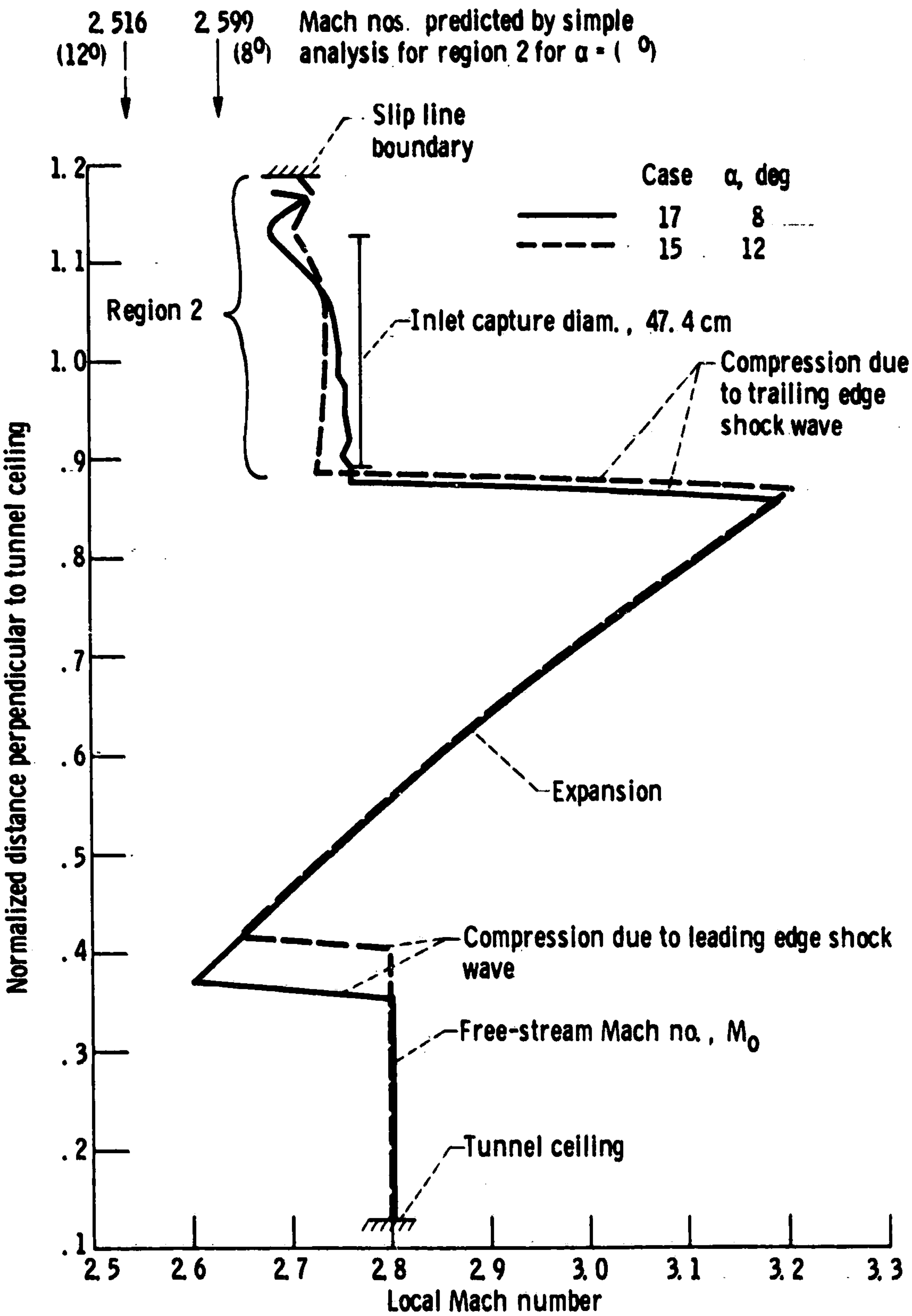


Figure 10. - Mach no. profiles predicted by detailed analysis at expected longitudinal location of inlet cowl lip. Triangular airfoil disturbance (see Fig. 7). Determination of disturbance region 2. Normalizing length, 204.5 cm.

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15. Supplementary Notes

16. Abstract

During the past several years, an attempt has been made at the Lewis Research Center to develop a device for perturbing the flow field in a supersonic wind tunnel. The goal of this work was to generate atmospheric type disturbances (e.g., gusts) and to investigate their effects on the dynamics and controls of supersonic inlets. Experimental data were also needed for verification and/or improvement of a NASA analysis of inlet dynamics for disturbances upstream of the normal shock. This report summarizes the status of development of a disturbance device including the desired aerodynamic and actuation capabilities of the device, and the techniques that have been considered and their drawbacks. At the present time no device has been found that satisfies the desired capabilities.

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